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RESEARCH MEMORANDUM

A DISCUSSION OF METHODS FOR REDUCING AERODYNAMIC HEATING IN SUPERSONIC FLIGHT

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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A DISCUSSION OF METHODS FOR REDUCING AERODYNAMIC HEATING IN SUPERSONIC FLIGHT

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INTRODUCTORY REMARKS

Because of the seriousness of the structural and other problems introduced by aerodynamic heating, considerable effort has been devoted to finding methods of reducing heat transfer in supersonic flight. It is the purpose of this paper to describe some of the more promising of these methods which have been discovered to date.

RESULTS AND DISCUSSION

Aerodynamic heating is brought about primarily by the convection of heat from the boundary layer to the surface of a vehicle. The severity of the heating is strongly dependent upon the flow in the boundary layer. To illustrate this point, consider flow of the familiar laminar and turbulent types (see, e.g., refs. 1, 2, and 3). The friction coefficients for laminar flow are usually substantially less than those for turbulent flow, and according to Reynolds analogy (see, e.g., refs. 4 and 5) the heat-transfer coefficients should be reduced in about the same proportion. The magnitude of this reduction is indicated in figure 1 where the Stanton number, which is proportional to the heat-transfer coefficient, is shown as a function of Mach number for laminar and turbulent flow over a flat surface at a Reynolds number of 10^7. The surface is presumed to be at ambient air temperature. Under these circumstances, it can be seen that laminar heat-transfer coefficients vary from about one-fifth the turbulent value at a Mach number of 2 to about one-third the turbulent value at $M = 8$. According to Newton's law of cooling, the heat-transfer rate per unit area is equal to the product of the heat-transfer coefficient and the difference between the recovery temperature (corresponding to zero heat flow) and the wall temperature. It follows that the laminar heat-transfer rates should be less than turbulent in about the same proportion as the heat-transfer coefficients inasmuch as the recovery temperatures are about the same for both types of boundary layer, and the wall temperatures are, of course, presumed unchanged. It is indicated then that both local and

over-all heating of a vehicle can be reduced by increasing the amount of laminar flow. This observation raises the fundamental question - how can the Reynolds number of transition from laminar to turbulent flow be increased?

Theoretical and experimental work to date (refs. 6 to 16) have indicated that there are three expecially promising methods of increasing the transition Reynolds number in supersonic flight: One method is to cool the surface of the vehicle; another is to shape the surface to give decreasing pressures with distance aft on the vehicle, that is, negative pressure gradients; and the third is to minimize surface roughness of the vehicle. The first two methods tend to stabilize the laminar boundary layer against disturbances which might otherwise cause transition to turbulent flow. The third method tends to eliminate one important source of these disturbances.

Some examples of the effects of surface temperature and pressure gradient on transition are shown in figure 2. First it is observed that cooling the surface from about 10 percent above to 10 percent below the recovery temperature increased the transition Reynolds number on a parabolic body of revolution from about 6×10^6 to 20×10^6 at a Mach number of 1.61 (ref. 12). In other words, the length of laminar run was increased by more than threefold. The effect of pressure gradient on transition Reynolds number (ref. 15) is also shown in this figure for the case of surface temperature equal to recovery temperature. The parabolic body, which has a negative pressure gradient for about three-fourths of its length, has about twice as long a laminar run as the ogive-cylinder which has a favorable gradient for only about the first quarter of its length. Similarly, the ogive-cylinder has about twice the laminar run of the cone-cylinder which has a negative pressure gradient only at its shoulder.

Pressure-gradient effects also become evident when cylindrical bodies fly at an angle of attack (ref. 11). This point is demonstrated in figure 3 where it is observed that the transition Reynolds numbers on the sheltered side of the body at an angle of attack are much smaller than those on the windward side. The effects shown can actually be correlated with a pressure rise coefficient along a streamline passing from the windward to the sheltered side of the body. More generally, it can be said that all the experimental effects shown here and in figure 2 agree qualitatively with the results of boundary-layer stability theory (refs. 6 to 9).

The effect of distributed surface roughness on transition Reynolds number (ref. 11) is shown in figure 4. Here the transition Reynolds number on a model fired in a free-flight wind tunnel is plotted against the ratio of roughness height to the laminar-boundary-layer thickness at transition. The roughness height is the depth of the screw thread used to create the roughness. It is apparent that increasing the roughness moves transition forward for each of the length Reynolds numbers at which tests were conducted.

There are, then, several methods which individually and collectively offer promise of reducing heat transfer in supersonic flight by maintaining more laminar flow over a vehicle. At best, however, there are numerous disturbances such as those produced by shock waves, noise, and wakes, which can still cause transition and which cannot be easily eliminated. Accordingly, it is appropriate to consider next a method which appears expecially suited to reducing heat transfer from a turbulent boundary layer to a surface. This method is termed "transpiration cooling" (see refs. 17 to 19).

With a transpiration cooling system, the coolant passes through the material to be cooled as shown in figure 5. The coolant may pass through as a gas, or as a liquid that would evaporate on the surface. A liquid has, of course, the advantage of absorbing the heat of vaporization during transpiration. In any event, the coolant leaves the surface as a gas and flows into the surrounding boundary layer. Naturally the outer skin of the aircraft would have to be porous. The porosity, however, introduces engineering problems of strength and of manufacture. Because of these and additional problems, consideration of transpiration cooling would only be made if it could be shown to be very effective. There are two reasons why a transpiration cooling system is thought to be effective. First, from a heat-exchanger viewpoint, a transpiration cooling system is efficient because it raises the temperature of the coolant to the temperature of the outer surface, where the highest temperature in the system exists. Thus, the coolant absorbs the maximum possible amount of heat. In addition, when the coolant leaves the surface as a gas, it reduces the shear in the boundary layer by tending to separate it from the wall. As a result, the heat transfer to the body is reduced. These two attributes, utilizing the coolant to its fullest and reducing the amount of heat entering the aircraft, make a transpiration cooling system exceptionally effective. Some indication of the reductions in heat transfer obtainable by this method is shown in figure 6 for the case of flow over a flat plate at a Mach number of 2.6 and a length Reynolds number of 5×10^6. The ordinate is the ratio of heat-transfer coefficient with, to heat-transfer coefficient without transpiration. The abscissa represents the ratio of mass flow of air per unit area through the surface to the mass flow per unit area of the free stream. The important conclusion to be drawn from this figure is that according to both theory (ref. 18) and experiment, large reductions in heat transfer are obtained by transpiring relatively small amounts of air.

Up to this point, we have concerned ourselves primarily with over-all heat transfer to a vehicle - local heat transfer has been discussed only in the sense that it contributes to over-all heat transfer. It is appropriate now to view the local heating problem as it relates to the generation of "hot spots" on a vehicle. In this regard, it will be undertaken to consider only those spots which are more or less common to all vehicles - namely, the nose of the body and the leading edge of the wing.

(There may, of course, be other regions of this type - for example, the transition region - however, it is beyond the scope of the present paper to consider these.)

A nose or leading edge tends to become excessively hot for the following reason. The boundary layer is just beginning to form so it is still very thin and the rate of viscous shearing is very high. Accordingly, the local heat-transfer coefficients are very high. In the event a nose or leading edge is sharp, it is indicated by boundary-layer theory that the local heat-transfer coefficients take on prohibitively large values. For this reason, and because a sharp nose or leading edge has little capacity for absorbing heat, effects of bluntness on heat transfer have been a matter of some interest (see, e.g., refs. 20 and 21). Theoretical studies of the heat transfer from a laminar boundary layer to a hemispherical nose or a semicircular leading edge indicate that, all other things being the same, the heat-transfer coefficients should vary inversely with the square root of the radius of curvature of the surface (see refs. 22 to 26). Figure 7 shows a comparison of this prediction with experimental data for transverse cylinders (ref. 26). The nominal test conditions were Mach number 9.8, stagnation temperature 2200° R, cylinder temperature 530° R, and cylinder Reynolds numbers from about 0.3×10^3 to 4.2×10^3. Variations in Re_D were obtained by varying cylinder diameter. For these tests, then, the dimensionless ordinate is proportional to heat-transfer coefficient while the abscissa is proportional to cylinder diameter. It is indicated that heat-transfer coefficients are substantially reduced by increasing the diameter, and essentially in the manner predicted by theory. Analogous results have been obtained for hemispherical noses (ref. 27). There is good reason to believe, then, that round noses and round leading edges are desirable from the standpoint of reducing the high rates of local heat transfer in these regions.

Now blunting the nose of a body in some cases actually reduces pressure drag (ref. 21). On the other hand, a substantial increase in pressure drag is almost inevitably associated with blunting the leading edge of a wing. Sweeping the leading edge is, of course, an effective means for minimizing this penalty.[1] The question is then raised, however, as to what effect sweep has on heat transfer. It was argued in reference 21 that sweep could be expected to markedly reduce heat transfer to a blunt leading edge at hypersonic speeds. The theoretical and experimental results of reference 26 substantiated this argument. Theory was worked out for the stagnation region in the limiting (and perhaps most practical) case of wall temperature that is low in comparison with stagnation temperature. Comparison of the predictions of this theory with experiments on heat transfer to swept cylinders is shown in figure 8. Specifically, there is shown the ratio of average heat-transfer rate at angle of sweep to

[1]It is pointed out in reference 21 that leading-edge pressure drag should, at hypersonic speeds, be reduced in proportion to the square of the cosine of the angle of sweep.

average heat-transfer rate at zero sweep as a function of the angle of sweep. The experimental data are for Mach numbers of 9.8 and 6.9 and wall to stagnation temperature ratios of 0.24 and 0.48 to 0.84.[2] The theoretical prediction is shown only for the temperature ratio of 0.24, inasmuch as the theory is not applicable at the higher temperature ratios. It is encouraging to note that according to both theory and experiment, large reductions in heat-transfer rates are achieved by sweep. Sweepback, then, may prove as useful in reducing heat transfer and drag in flight at high supersonic speeds as it has been in reducing drag at low supersonic speeds.

So far in this discussion it has been presumed that the boundary layer is normal in the sense that it is laminar, transitional, or turbulent, and more or less steady. As a final point, however, we depart from this presumption and consider a basically different flow (ref. 28) which offers promise of transferring less heat to a surface than the usual boundary layer. This flow is nonsteady, rather than steady, and it may be represented schematically as shown in figure 9. It differs from the normal boundary layer in one important respect - large vortices exist in the boundary layer at short intervals along the surface. Now, assume that each vortex is made up of air from the main flow and, further, that the surface is at some reasonable temperature greater than ambient air temperature. It may be argued then that part of the heat convected to the surface in the region of the normal boundary layer should be convected from the surface in the region of a vortex since the peripheral portion of the vortex should be cooler than the surface. The cooling by the vortices should, of course, depend upon both their size and spacing.

It was undertaken to check the principle of this "boundary-layer-vortex" hypothesis using specially designed spike-nosed bodies of revolution. These bodies were employed simply because they are known to generate a high-frequency pulsating flow at supersonic speeds (ref. 29), and it seemed reasonable to expect that large-scale vortices would be discharged over the bodies with each pulsation. Visual flow studies (ref. 28) indicated that this was, in fact, the case. It was anticipated further that aerodynamic cooling by the vortices would reflect directly in the recovery temperatures of the surface. Some idea of the magnitude of this effect at zero angle of attack and a Mach number of 3.5 can be obtained from figure 10. Here the recovery factor at a representative point on the surface of a spike-nosed body is shown as a function of spike extension. The body consisted of a truncated cone with a conical spike and an annular cutout added to the front face of the cone. The purpose of this cutout is to produce stronger, better developed vortices during pulsation. For spike extensions up to about 0.4 inch the flow remained steady and the recovery factor remained at a rather high value, characteristic of turbulent boundary layers. With further spike extension,

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[2]The data for a Mach number of 9.8 are from the Ames hypersonic gun tunnel (ref. 26) while those for a Mach number of 6.9 are from the Langley 11-inch hypersonic wind tunnel.

pulsating flow occurred at the nose, and the bow shock oscillated between the front face of the model and the tip of the spike. Ring-shaped vortices were discharged over the model in concert with each pulsation and, as can be seen, substantial reductions in recovery factor were obtained at spike extensions up to 2.2 inches. With further spike extension the pulsations tended to disappear, large vortices were no longer shed from the nose, and the recovery factor rose to values in the range of normal boundary layers. The maximum reduction in recovery factor due to spike extension was from about 0.91 to 0.68.

It is of interest now to see what the distribution of recovery factor is like on the spike-nosed body under conditions of pulsating flow. Figure 11 provides some information on this matter for a spike extension equal to about three-fourths of that for obtaining minimum recovery factors. It is observed first that the recovery factors rise from a relatively low value near the center of the spike to a high value near the annular cutout. On the afterbody something like the proposed boundary-layer-vortex flow was obtained and it is observed that the recovery factors are uniformly low, varying from about 0.72 to 0.75. These results and those of the previous figure indicate that recovery temperatures on the surface of a vehicle in flight at a Mach number of 3.5 in air at 40° F ambient temperature could be reduced from about 1150° F to as low as 870° F by using vortex cooling. Large vortices offer promise, then, of substantially reducing recovery temperatures in flight at high supersonic speeds.

It is appropriate to inquire next as to what effect the pulsating flow has on heat-transfer coefficients. Some idea of this effect can be obtained from figure 12 which shows indicated average heat-transfer coefficients as a function of the difference between indicated recovery and wall temperatures of a spike-nosed model and a truncated cone model. Each model was solid and made of duralumin. The term "indicated" is used because the temperatures were measured at the base of the models. There are two important things to be noted. First, the effect of pulsating flow on heat-transfer coefficients is rather small, in this case reducing the coefficients by perhaps 7 percent. Second, the heat-transfer coefficients for the pulsating flow are, as in the case of steady flow, more or less constant, independent of wall temperature.

The latter observation suggests that the measured recovery factors can be combined with the measured heat-transfer coefficients in order to estimate the effects of pulsating flow on average heat-transfer rates in flight. It seems unlikely that in practice these rates would exceed those corresponding to the case of wall temperature equal to ambient air temperature. For this particular case, the rate of heat transfer is simply proportional to the product of the heat-transfer coefficient and the recovery factor; but pulsating flow reduced the heat-transfer coefficient by about 7 percent and it reduced the recovery factor from 0.91 to about 0.77, or about 16 percent. The net reduction in maximum average heat-transfer rate is indicated, then, to be about 23 percent. For heat-transfer rates less

than the maximum, the percentage reduction associated with pulsing vortices over a body should be greater because the role played by recovery factor is of increased importance.

CONCLUDING REMARKS

In the light of the previous considerations, the following conclusions are reached regarding presently available methods of reducing aerodynamic heating in supersonic flight. First, since a laminar boundary layer convects heat to a surface much less rapidly than a turbulent boundary layer, it appears worthwhile to strive for larger amounts of laminar flow by employing smooth, cooled surfaces which are shaped to give favorable negative pressure gradients. In the event the turbulent boundary layer cannot be avoided, the resulting aerodynamic heating may be substantially reduced by using a transpiration cooling system. Excessive heating of the noses of bodies and leading edges of wings can be minimized by blunting or rounding these surfaces, and in the case of wings this local heating problem can be further alleviated by sweeping the leading edge. Finally, there is the encouraging possibility of reducing aerodynamic heating below that encountered with normal boundary-layer flows by employing vortex cooling.

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National Advisory Committee for Aeronautics
Moffett Field, Calif., June 21, 1955

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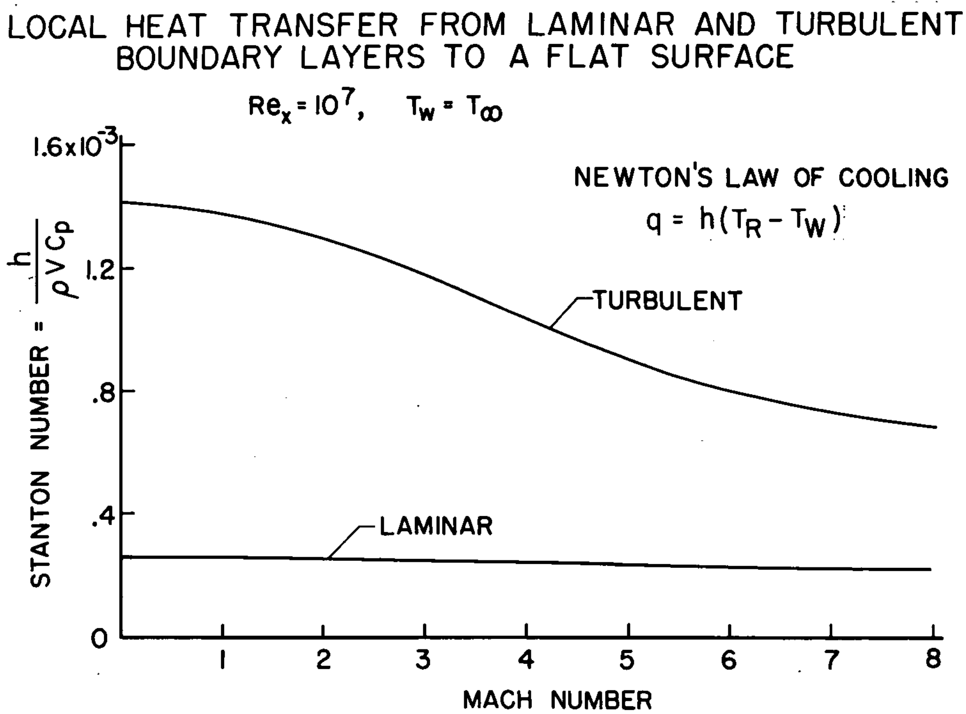

Figure 1

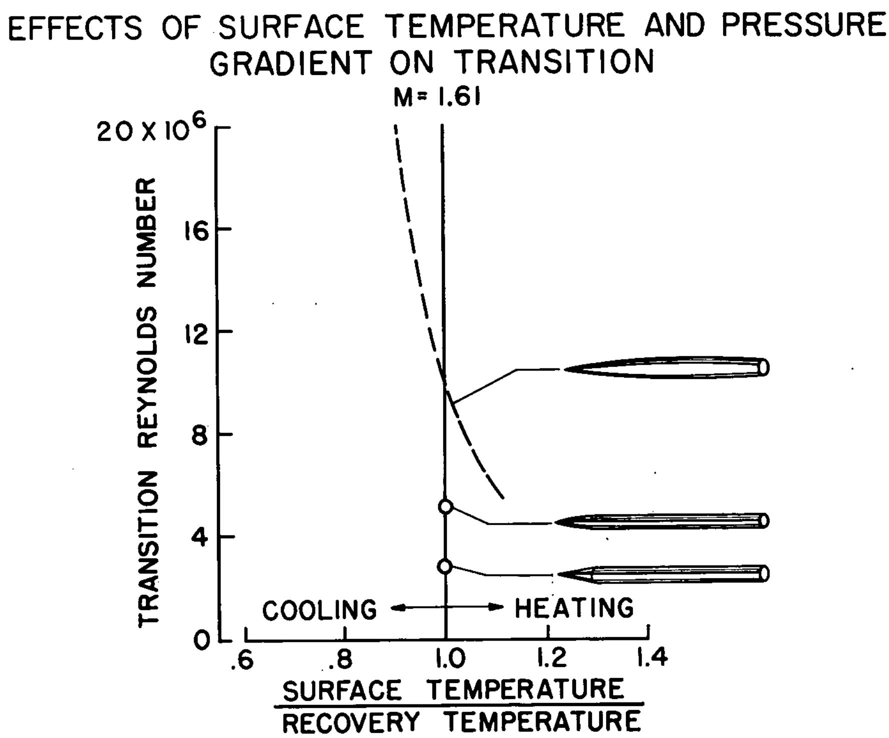

Figure 2

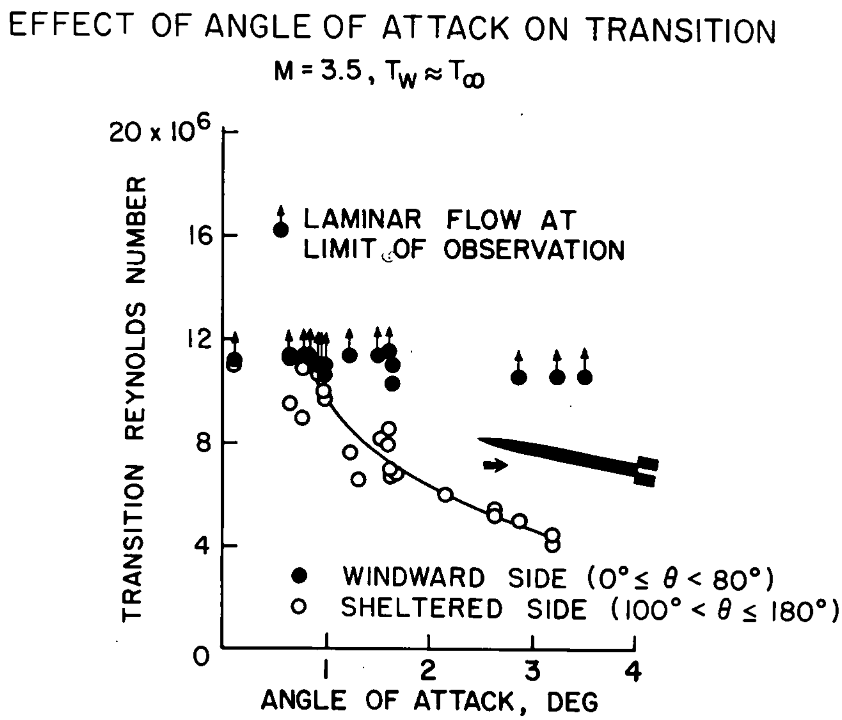

Figure 3

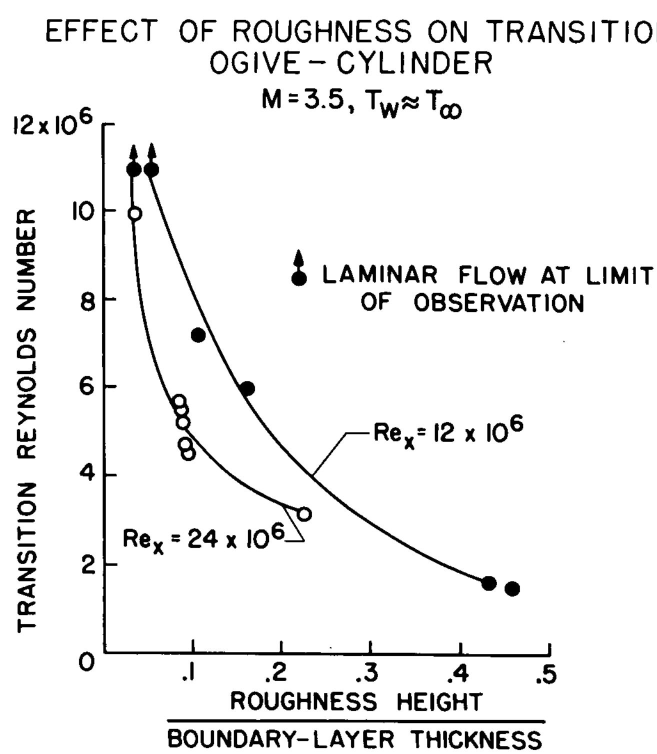

Figure 4

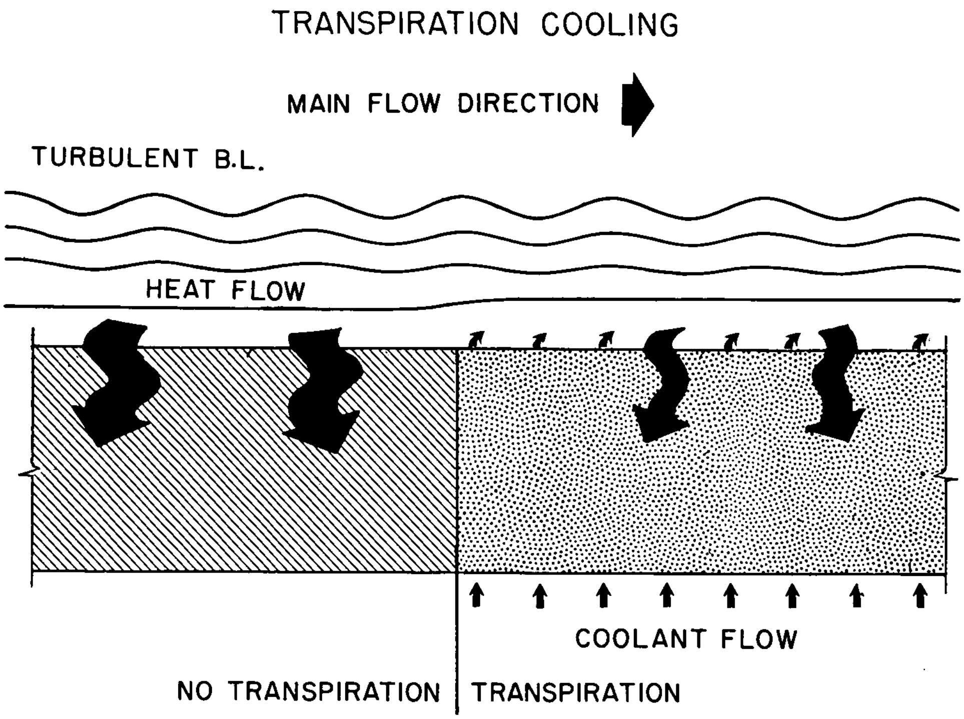

Figure 5

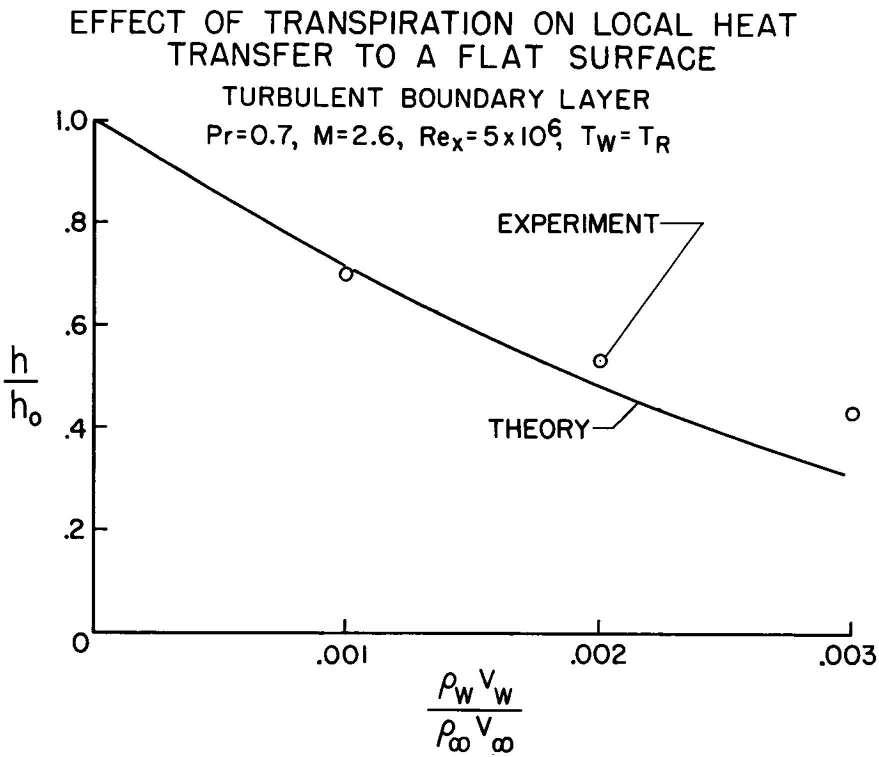

Figure 6

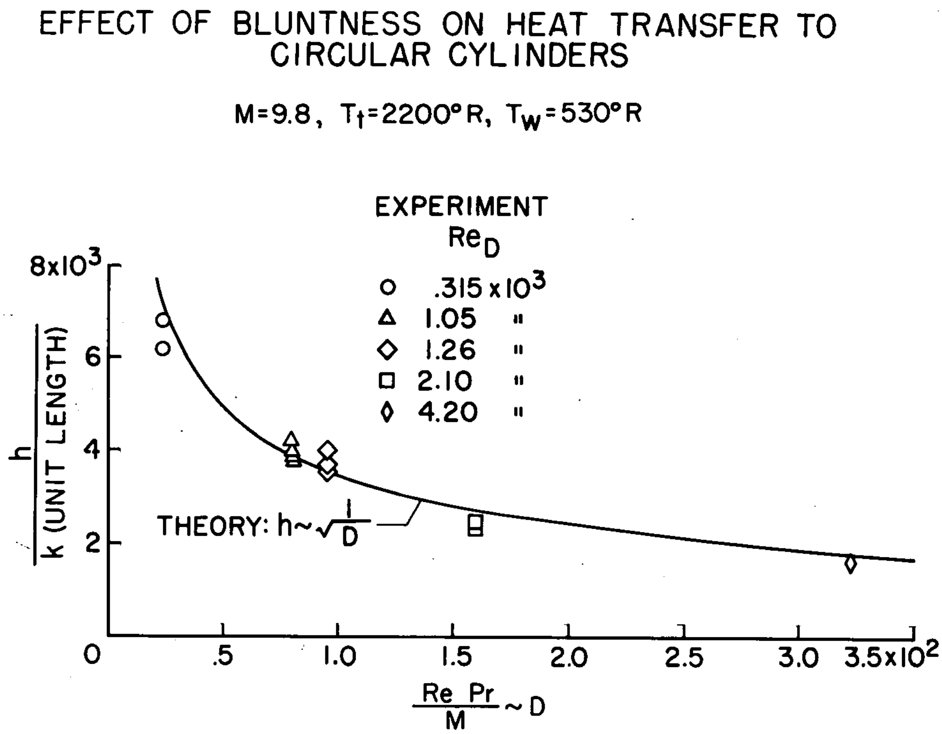

Figure 7

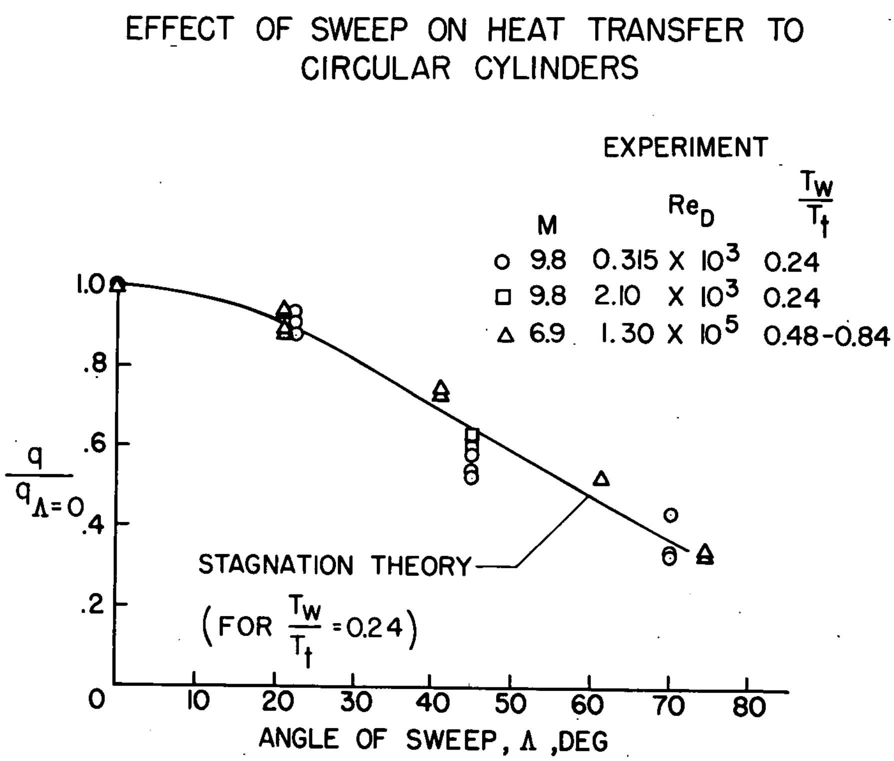

Figure 8

MODEL OF BOUNDARY-LAYER-VORTEX FLOW

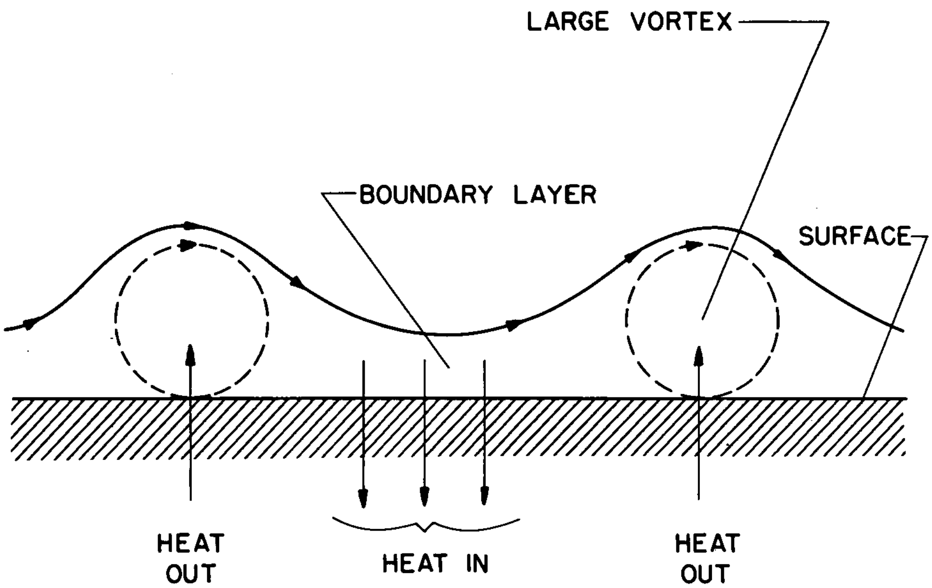

Figure 9

EFFECT OF SPIKE EXTENSION ON RECOVERY FACTOR

$Re_D = 1.9 \times 10^6, \alpha = 0°, M = 3.5$

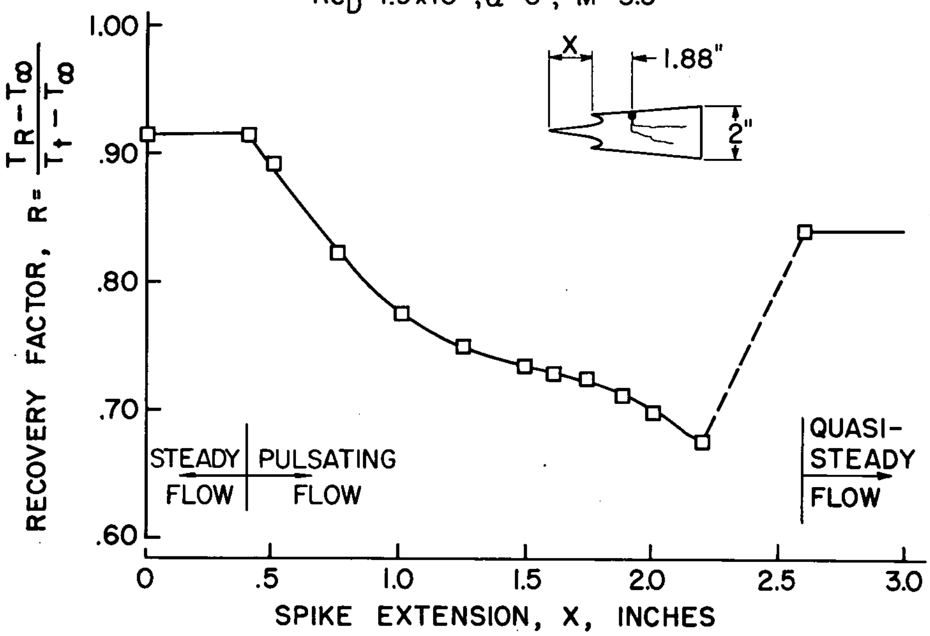

Figure 10

RECOVERY FACTORS ON SURFACE IN PULSATING FLOW

$\alpha = 0°$, M = 3.5

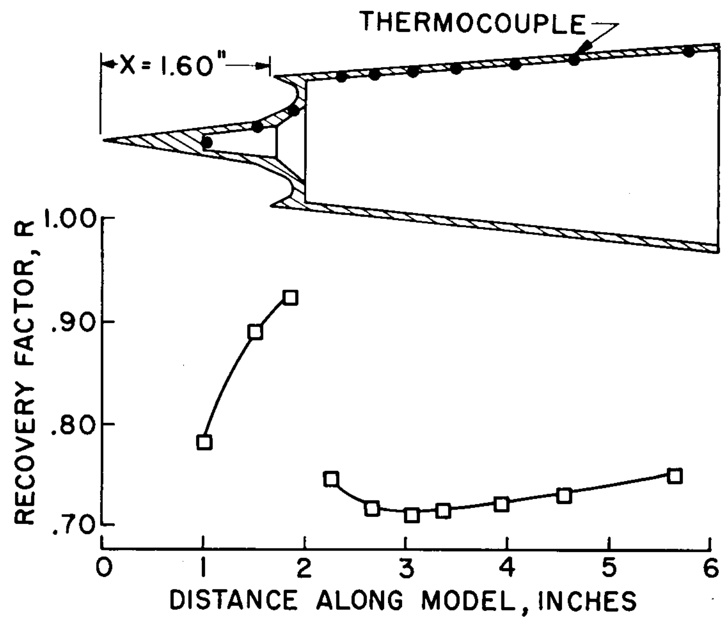

Figure 11

EFFECT OF PULSATING FLOW ON HEAT-TRANSFER COEFFICIENTS

$\alpha = 0°$, M = 3.5

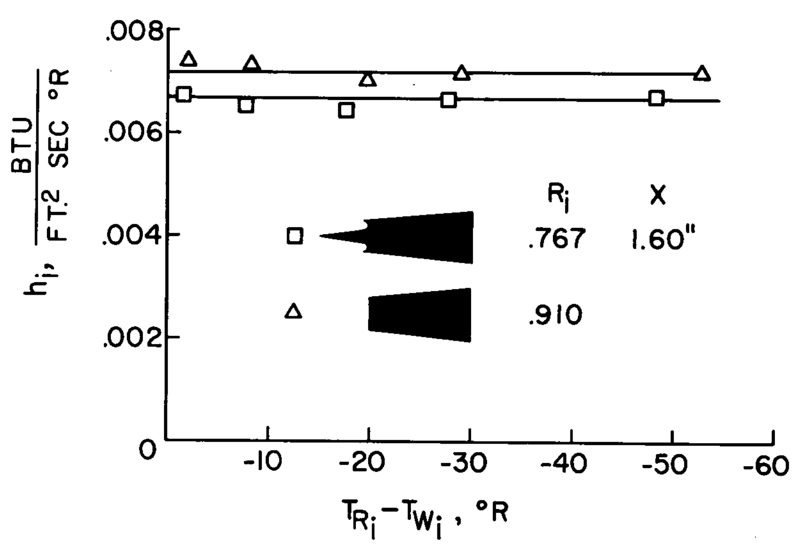

Figure 12